A Fractionated Space Weather Base at L_5 using CubeSats and Solar Sails

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The Sun-Earth L_5 Lagrange point is an ideal location for an operational space weather forecasting mission to provide early warning of Earth-directed solar storms (coronal mass ejections, shocks and associated solar energetic particles). Such storms can cause damage to power grids, spacecraft, communications systems and astronauts, but these effects can be mitigated if early warning is received. Space weather missions at L_5 have been proposed using conventional spacecraft and chemical propulsion at costs of hundreds of millions of dollars. Here we describe a mission concept that could accomplish many of the goals at a much lower cost by dividing the payload among a cluster of interplanetary CubeSats that reach orbits around L_5 using solar sails.

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I. Introduction

The ascendancy of CubeSats has brought renewed interest in solar sail propulsion for the simple reason that commercially available sails are large enough to propel low mass CubeSats over interplanetary distances in reasonable amounts of time. Here, we present a concept for a space weather forecasting base to provide early warning of approaching solar storms utilizing a loose cluster of interplanetary CubeSats that reach orbits around the Sun-Earth L_5 Lagrange point (Fig. 1) using solar sails. The most destructive solar storms result from fast coronal mass ejections (CMEs) - enormous clouds of plasma and magnetic field explosively ejected from the corona at speeds reaching 2000 km/s. The release of these CMEs is often accompanied by a large solar flare that sends relativistic particles streaming into space. The fast CMEs drive shock waves that accelerate large fluxes of energetic particles. These solar energetic particles are hazardous to spacecraft and humans in space. CMEs impacting Earth's magnetosphere can cause geomagnetic storms that can damage power grids and communications systems. By placing appropriately instrumented spacecraft at L_5, solar storms heading towards Earth can be detected in time to give early warning so the effects can be mitigated. The major advantages of the L5 position for forecasting are that it provides a view of the inner heliosphere to image CMEs one to five days before they reach Earth and it allows viewing of solar active regions behind the solar limb that are rotating Earthward. Both provide greater warning time than simply having sensors at Sun-Earth L1. The insert in Fig. 1 shows an Earth-directed CME imaged from one of NASA's Solar TErrestrial RElations Observatory (STEREO) twin spacecraft.

The CubeSat mission presented here draws heavily on a NASA Innovative Advanced Concepts (NIAC) study [1] that developed a concept for a small spacecraft for interplanetary missions with a target volume of 10 cm x 20 cm x 30 cm (6U in CubeSat parlance, where 1 U = 10 cm x 10 cm x 10 cm). This study allocated 2U for a solar sail; the sail system was based on the Planetary Society's LightSail-1™ architecture (www.planetary.org/explore/projects/lightsail-solar-sailing/). The concept also draws on two other activities: (1) the unsuccessful SolWISE (Sailing On Light With Interplanetary Science and Exploration) 6U CubeSat proposal, led by A. Klesh, submitted to NASA's Edison call in 2012; and (2) the proposed 3U INSPIRE (Interplanetary NanoSpacecraft Pathfinder In a Relevant Environment) interplanetary CubeSat mission, also led by A. Klesh, now under development at JPL, which has been selected by NASA's CubeSat Launch Initiative for an upcoming launch, possibly as soon as 2014.

The "fractionated" L_5 mission concept was developed at a 2012 workshop entitled "Small Satellites: A Revolution in Space Science," hosted by the Keck Institute for Space Studies at the California Institute of

Technology (http://www.kiss.caltech.edu/study/smallsat/). The concept would enable a permanent space weather forecasting base at L_5 that could accomplish, at a much reduced cost, the goals of a conventional single-spacecraft L_5 mission, as described in the recent heliophysics decadal report [2]. Rather than using a single conventional spacecraft, multiple small satellites would be used with the scientific payload divided among them; thus the name fractionated Space Weather Base (SWB-L5). Key to the concept is that only one of the CubeSats would carry a high-gain antenna and other hardware necessary for sending high-rate science data to Earth (~1 AU from L_5). The other spacecraft would carry a much smaller communication system to send the science data to the communication hub and low-rate engineering data to Earth.

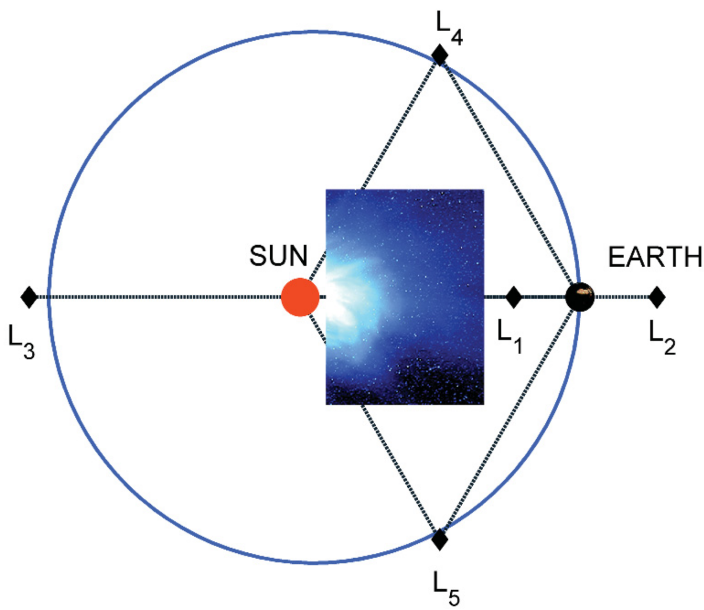

Fig. 1 Schematic showing Earth – L5 - Sun relative locations. Insert shows a white light image of a CME taken from an Heliospheric Imager on one of NASA's STEREO spacecraft.

For the first phase of operation, a loose (~1000 km maximum separation) constellation of five 6U CubeSats would be placed at L_5. Each of the ~10 kg CubeSat allocates ~2U for the solar sail, ~2U for the engineering subsystems common to all five constellation members (attitude control, avionics, etc.) and ~2U for each constellation member's unique payload. This allocation of space is based on the architecture developed for the SolWISE proposal, shown in Figs. 2 and 3 (deployment sequence). The SWB-L5 mission could later be expanded incrementally to add new instruments and new objectives by sending additional small spacecraft to the L_5 base. The mission concept described in this paper represents a potential beginning for a permanent space weather forecasting base at L_5.

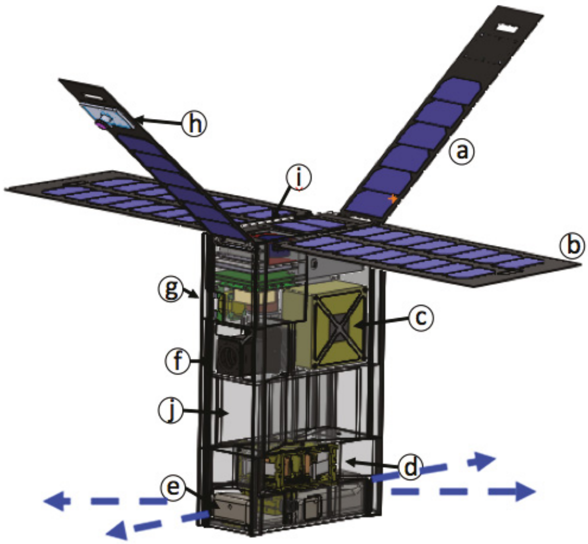

Fig. 2 The 6U SolWISE CubeSat shown after sail deployment (with sail, booms and sides hidden). The booms deploy in the direction of the blue arrows. (a) Fixed solar panels, (b) Rotatable solar arrays, (c) Attitude determination and control system, (d) Sail deployer, (e) Cold-gas system, (f) Star tracker, (g) Magnetometer (stowed), (h) Camera (to observe sail deployment), (i) Solar wind instrument, (j) Sail cavity. The SWB-L5 spacecraft could have a similar architecture.

While detailed engineering studies of the fractionated mission concept have not yet been done, the approach has several obvious advantages and cost savings: (1) Existing solar sails are sufficient for propulsion: the trajectory calculation described in Section II shows that a ~6U spacecraft with a sail approximately twice the Lightsail-1™ size (using thinner sail material and stronger material in the sail boom support structure, at somewhat higher cost than the lowest-possible-cost Lightsail-1™ implementation) could reach an orbit around L_5 in less than 3 years; (2) Spacecraft requirements are eased when the *in situ* instruments are not on the same spacecraft as the imaging instruments: fields and particle instruments prefer spinning spacecraft whereas imaging instruments require spacecraft with 3-axis stabilization, and often instruments are susceptible to interference from each other; (3) Integration and testing is much easier and cheaper for several simple small spacecraft than one large spacecraft with many instruments with conflicting requirements; (4) The cluster could be built up incrementally; (5) Fractionation allows incremental replacement of degrading or failed spacecraft at dramatically lower cost than if a single spacecraft is utilized to provide full capability; (6) Different agencies or institutions could contribute their own CubeSat; (7) Individual CubeSats could be replaced to upgrade capabilities; and (8) The Space Weather Base could be expanded later by adding other (perhaps larger) spacecraft with new instruments (solar coronagraph, solar EUV and X-ray imagers and spectrometers) to address additional science goals.

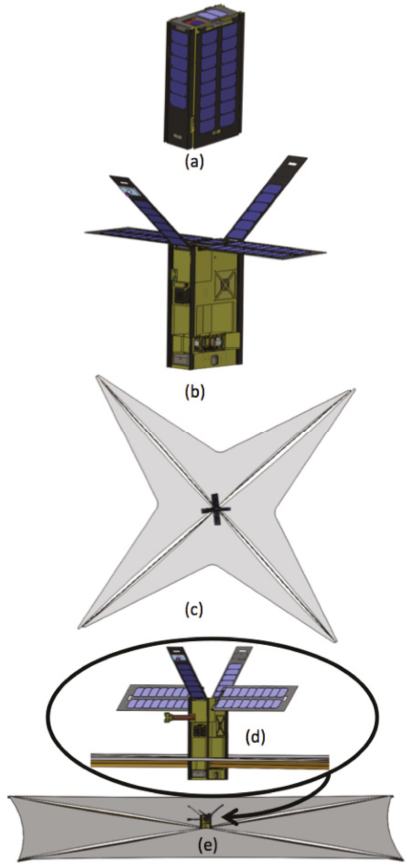

Fig. 3 SolWISE deployment sequence with precisely controllable and monitored elements: a) stowed spacecraft, b) solar panel deployment, c) boom and sail deployment, d) magnetometer deployment, e) fully deployed spacecraft. The SWB-L5 spacecraft could have a similar deployment sequence.

In Section II, calculation of the trajectory to L_5 and the constellation formation are discussed. Section III contains an overview of the fractionated mission, including the science goals, the division of the payload among the five constellation members and potential expansion of the concept beyond the first phase. In Section IV, the science and engineering subsystems of the spacecraft are described and the technology challenges discussed. Section V contains a summary and discussion.

II. Solar Sail Trajectory Calculation

Since a ~10 kg 6U CubeSat cannot carry enough propellant to reach L_5 using conventional propulsion, a solar sail is used. Each of the constellation members would transfer directly from an Earth-escape orbit into an orbit around L_5 using its own solar sail. The spacecraft may be launched on a single or multiple launches as secondary payloads at very low cost. The sail system itself is described in Section III. For this mission, it was desired that the final orbit about L_5 be relatively small (~1000 km maximum separation of constellation members) to ease the requirements on the inter-CubeSat communication system. The SWB-L5 trajectory results are generated for a solar sail design with characteristic acceleration (the acceleration at 1 AU with sail facing the Sun) a_0=0.068 mm/s^2 and 100% efficiency, equivalent to a perfectly reflecting 8.7 m square sail with total mass of 10 kg. This assumption is actually somewhat conservative given that the supplier has stated their ability to provide a sail up to 10 m on a side within the allocated stowage and deployer volume as soon as 2015.[13]

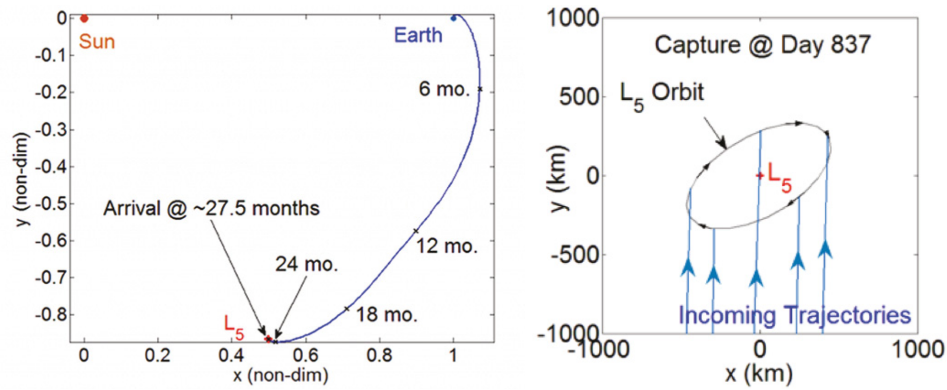

Fig. 4 (a) Overview of solar sail trajectory from an Earth-escape orbit to the ~1000 km diameter orbit around L_5 orbit, shown in SEMB PCR3BP rotating frame (see text). Travel time to L_5 is 837 days. (b) Close up of the L_5 orbit, showing points of insertion of the five trajectories. Orbits were calculated to place five spacecraft with equal temporal spacing in final orbit to show ease of assembly of the constellation.

In order to transfer to the ~1000 km diameter L_5 orbit directly starting from an Earth-escape orbit (characteristic energy $C_3 \approx 0$ km^2/s^2) using solar sail propulsion, a backward integration scheme is used. The initial state condition of the backward integration is set to coincide with a chosen location on the periodic L_5 orbit so as to constrain the transfer trajectory to end in the desired state around L_5. Thereafter a simple single-parameter control law, described below, is used to propagate the trajectory backward from L_5 and, by varying the one parameter, find a ~200 km altitude flyby of Earth with $C_3 \approx 0$. One such trajectory is shown in Fig. 4a. The reason for starting from an Earth Escape trajectory is because the Earth's gravity well is so deep, that our current

[13] Tomas Svitek, CEO, Stellar Exploration Inc. personal communication to Robert L. Staehle, 2013 April 16.

sail design would require over two years to escape the Earth starting from geostationary Earth orbit (GEO). To eliminate this long escape trajectory, we assumed the launch vehicle would deliver our CubeSats into an Earth-escape orbit with a perigee altitude of ~200 km and $C_3 \approx 0$. Upper stage disposal to escape is not unusual, having been used most recently on the Landsat 8 launch in 2013 February. There is often excess payload capability within which one or more CubeSats can be accommodated. Deployment using excess payload capability from GEO satellites[14] remains a backup option. Such secondary launch opportunities may be available for costs on the order of $1 M for a 6U CubeSat.

The underlying dynamical equations are the Planar Circular Restricted 3 Body Problem (PCR3BP) with the Sun (S) and Earth-Moon Barycenter (EMB) as primary and secondary bodies respectively, referred to as SEMB PCR3BP hereafter [3]. Acceleration due to solar radiation pressure is also modeled, where a perfectly reflective solar sail is assumed. The equations of motion are given in [4]. The control is the angle of the sail normal with respect to the Sun-Spacecraft line. The control law chosen to achieve the transfer is based on the idea that an Earth to L_5 transfer is similar to a heliocentric co-orbital rendezvous to a point trailing Earth by approximately 60°. Thus, one must first increase the orbital specific energy to slow down with respect to Earth and thereafter decrease the energy to speed up and re-achieve orbital velocity to remain at the target once it is reached. An approximation of a locally optimal sail steering law was used for maximizing and minimizing the orbital energy rate of change. For a sail in a circular heliocentric orbit in the 2-Body Problem, the sail angle that maximizes the rate of heliocentric orbital energy increase is approximately 35°, derived by McInnes[5]. Conversely, the sail angle that maximizes the rate of subtracting energy from the heliocentric orbit is approximately -35°. The parameterized control law is set up to use the maximizing sail angle while leaving Earth and switches over to the minimizing sail angle once a specified "mean anomaly" is achieved, where the mean anomaly is measured by the angle between the X-axis of the SEMB PCR3BP coordinate system and the spacecraft position vector (see Fig. 4a, which shows a trajectory in a frame rotating with the center of mass of the Sun-Earth-Moon system (SEMB PCR3BP rotating frame)**.** This parameter, the switching mean anomaly, is the single control parameter that was manually modified by trial and error to achieve an Earth flyby at a desired ~200 km altitude with $C_3 \approx 0$. The selection of a 200 km perigee for this example is merely a convenience used in order to estimate approximate flight time; in reality a likely starting orbit would have a higher perigee but still have $C_3 \approx 0$. Flight time from Earth's vicinity to L_5 is relatively insensitive to this perigee parameter.

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[14] David Lackner, Space Systems Loral, private communication to Robert Staehle, 2011 June 23. Published costs are also available from Spaceflight Services online (www.spaceflightservices.com).

The initial guess for the switching mean anomaly was -30° from Earth, since this is halfway to the destination. Thus the spacecraft is accelerating for the first half and decelerating for the second half. Five individual trajectories were generated from an ~200 km altitude flyby to the same (periodic) L_5 orbit at five positions, equally spaced in time on the periodic L_5 orbit (See Fig. 4b). This was done to illustrate that a constellation could be placed in orbit at L_5 with arbitrary spacing requirements through use of solar sail propulsion. Since the concept is to incrementally build the constellation and to continually add to the constellation, the number of constellation members could vary substantially. For this paper, we used five spacecraft to demonstrate the feasibility and ease with which such constellations may be assembled and controlled at L_5. All resulting trajectories are slightly different but have an approximate transfer time of 837 days. The Earth flyby altitudes vary in the range 143-270 km. At 90% efficiency, the transfer time is 961 days. These results are summarized in Fig. 4. Fig. 4a shows a trajectory from Earth to L_5 and Fig. 4b shows the final periodic orbit around L_5 and the locations of the five trajectories with equal temporal spacing; both are in the SEMB PCR3BP rotating frame. Orbits about L_5 are operationally stable, so very little station keeping should be required.

In the above calculations, it was assumed that the spacecraft had a mass of 10 kg, but there is no requirement that the mass of each constellation member be exactly the same. Fig. 5 illustrates the relationship between spacecraft mass and sail size for this specific sail performance (characteristic acceleration a_0=0.068 mm/s^2 and 100% efficiency, where a_0= 8.22 / σ and σ is the sail areal density in g/m^2, which includes the support structure).

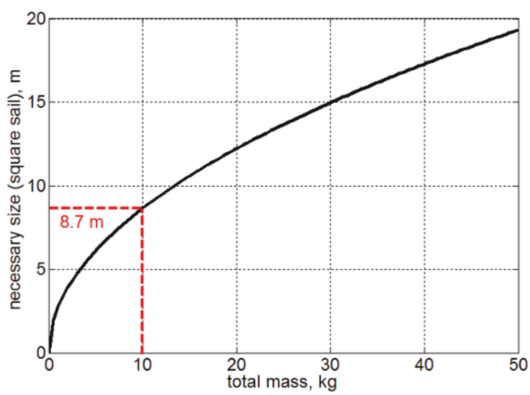

Fig. 5 Sail size vs. mass relationship for a sailcraft with a characteristic acceleration of a_0=0.068 mm/s^2. A perfectly reflecting square sail is assumed.

III. Mission and Spacecraft Overview

The L_5 point has been identified in the NRC Solar and Heliospheric Panel report as a prime location for a space weather forecasting mission for several reasons: (1) it provides global coverage of the inner heliosphere with the potential for observing CMEs and other transient disturbances one to five days before they reach Earth; (2) it allows viewing of solar active regions behind the solar limb rotating Earthward, and (3) it allows *in situ* sampling of solar wind structure at a heliographic longitude distinct Earth's and rotating Earthward, giving ~5 days warning. An L_5 mission would build upon experience using NASA's STEREO mission that had twin spacecraft, carrying both *in situ* and remote sensing instruments, drifting slowly away from Earth at 1 AU. The twin STEREO spacecraft did not have adequate propulsion to stop their drift away from Earth at L4 or L5. This illuminates one advantage of using propellantless solar sails; they can trade time for a velocity change ΔV, instead of trading limited mass for limited ΔV.

The minimal instrumentation necessary to accomplish the above goals is (1) a heliospheric imager to observe and predict trajectories of CMEs that might impact Earth; (2) a magnetograph to measure magnetic active regions on the Sun behind the solar limb which are rotating Earthward (3) an *in situ* solar wind package with a magnetometer to measure the solar wind magnetic field and instruments to measure the solar wind plasma temperature, velocity, density and composition; and (4) an energetic particle instrument to warn of solar energetic particle conditions in the solar wind rotating Earthward. This is the payload that would be carried in the first phase of SWB-L5 mission, divided among the five 6U CubeSats of this fractionated space weather forecasting base. Each of the constellation members would carry a unique payload, fitting in ~2U of the 6U CubeSat, as well as the required engineering subsystems that would be nearly identical for the five CubeSats.

A. Common Engineering Subsystems (~4U)

Each spacecraft would carry a solar sail system requiring two units (2U) of the 6U sailcraft. The remaining basic engineering subsystems, also common to all five, would require an additional 2U. These are (1) a communications system for inter-spacecraft communication and for sending very low rate engineering data to Earth, (2) an attitude determination and control system which would include a star tracker, cold gas thrusters, low-quality gyroscope, and basic Sun sensors, (3) avionics which would include the primary computer, power regulator, battery, onboard memory and a fault tolerant watchdog system, and (4) solar panel system (solar panels and deployment mechanisms). These systems and their current level of technology development for CubeSats are described in the next section.

B. Unique Payloads (~2U)

The unique payloads of the five sailcraft, each require the remaining 2U, are listed here. These subsystems and their current level of technology development are also described in the next section.

Spacecraft 1: High Gain Antenna (HGA) and other hardware necessary to collect science data (from the other constellation members) and communicate the data to Earth. This spacecraft is referred to as the communications hub (3-axis stabilized).

Spacecraft 2: A white light telescope (Heliospheric Imager) to image coronal mass ejections (3-axis stabilized).

Spacecraft 3: A magnetograph to measure the magnetic fields on the surface of the Sun (3-axis stabilized).

Spacecraft 4: A solar wind plasma instrument and magnetometer to measure the interplanetary magnetic fields (spin stabilized, after arriving on-station in L5 orbit).

Spacecraft 5: An energetic particles instrument and magnetometer to characterize the solar energetic particles (spin stabilized, after arriving on-station in L5 orbit).

The L_5 mission envisioned in [2] included additional instruments, to image the Sun and corona in white light, EUV and X-rays, that are unlikely to fit in the available 2U payload volume and that also require larger data rates than the instruments listed above. These instruments increase our understanding of the link between solar magnetic activity and its consequences in the heliosphere and would help develop the capability for even longer term space weather forecasting, e.g. the ability to predict which active regions are about to spawn a CME or flare. These other instruments could be included on larger spacecraft which would travel to L_5 by more traditional methods, or efforts could be undertaken to reduce the size and power requirements of these instruments to fit within the available CubeSat volume, if possible.

IV. Engineering and Science Subsystems

CubeSat capabilities have advanced rapidly as the community has matured over the last ten years. Today, attitude control systems are commercially available providing degree and sub-degree accuracy, power systems can provide 30+W of power, and onboard processing systems include FPGAs (Field-Programmable Gate Array), ARM [Advanced RISC (Reduced Instruction Set Computing) Machine] and other microcontroller architectures. A survey of available technologies can be found in [6,7]. But a unique aspect of CubeSats is that they are constrained by volume due to the launch system. Thus, though many individual capabilities are available through subsystems, not all the subsystems at the "bleeding edge" of capability, can fit within the same spacecraft. Yet thinking “inside the box" has led to major improvements in design that has continued to increase the overall spacecraft functionality without increasing size. 1U, 3U and 6U platforms are now standardly available, the shapes being dictated primarily by the launch availability.

Below is an overview of each subsystem needed for the SWB-L5 mission and its current state of the art:

A. Engineering Subsystems

Solar Sail Subsystem: The SWB-L5 solar sail system would be similar to that developed for the SolWISE proposal. For SolWISE, the solar sail deployment mechanism, booms and sail were scaled up versions of the existing LightSail-1™ design, built by Stellar Exploration (see http://www.stellar-exploration.com/#!lightsail). These booms are ~3 m longer with upgraded material and the deployer spindle is wider to accommodate them. The full spacecraft areal density would be 134 g/m^2. The sail material would be 5.5 μm Kapton™-E, which is available from DuPont. Kapton™ is also the material used by the Japanese solar sail mission IKAROS for its multi-year mission. Kapton™ is well known for its strong performance in space applications: testing at NASA MSFC showed it vastly out-endures Mylar™, the material used by NASA Ames and Marshall to construct NanoSail-D [8]. The sail would be covered by a thin layer of Al (nominal reflectivity ~90%) and reinforced with a 0.5" x 0.5" grid of 67-denier polyester yarn, providing a strong rip-stop material in case of damage. A reinforced, doubled-over edge, along with grommets at the corners, would assure sail strength during deployment. The sail is grounded to the spacecraft through the metallic booms, minimizing the potential for charge build-up. Due to the high-efficiency Z-fold packing, tested with both Kapton™ and Mylar™, the sail can significantly scale in size, limited only by the packed volume (~2U equivalent volume for SolWISE; somewhat less for LightSail™-1) and boom length. The sail deployment was shown in Fig. 3.

Avionics: Today, CubeSat avionics are generally commercially available radiation-soft hardware that is commonly found in hobbyist hardware or smart-phones. Simple, very cheap microcontrollers, advanced RISC Machine (ARM) processors, and even FPGA’s have served as the central processing core of CubeSats. As a common theme, each system is designed to expect resets; many systems have cascaded watchdog controllers to monitor and reset the system if latch-up occurs. Other systems include hardware timers that will reset the spacecraft automatically after a set period (typically a few days). These approaches allow latch-up robustness to be added to commercial processors. As the community matures, more information will become available regarding the performance of these controllers on orbit, and particularly, how well they have survive ionizing radiation (whose rate is higher in low Earth orbit than in the deep space environment of L_5). The mass of an avionics processor is typically ~100g and a single 10x10 cm printed circuit board in size. More advanced avionics boards often include memory storage, real-time clocks, telemetry sensors, and basic attitude determination devices (such as a micro-electro-machined gyroscope). INSPIRE, which may launch as early as 2014, would use a slightly modified version of the Radio Aurora Explorer (RAX-2) avionics boards, developed at the University of Michigan (http://rax.engin.umich.edu/). For the longer duration L_5 mission, it would be desirable to have more radiation tolerant boards. One example is JPL's CubeSat Onboard Processor Validation Experiment (COVE) flown aboard the University of Michigan's M-Cubed spacecraft [9], with COVE-2 for slated flight later in 2013. COVE has at its heart the rad-hard by design Xylinx Virtex-5QV FPGA. This is a rapidly evolving area and it can be expected that more fault tolerant boards will be available in a few years.

Electrical Power System and Battery: The electrical power system can be viewed as the core of the spacecraft. As CubeSats have limited surface area for solar cells, regulators are often needed to buck or boost the voltage to appropriate levels for distribution to systems, and for battery charging. Input regulators are often simple direct-energy circuits, but have been advanced to include single-set-point controllers, or even maximum power point trackers. Onboard batteries are typically lithium-ion cells, with a total energy around 5-10 A-hrs (for a 6U). The batteries are often the densest equipment aboard the spacecraft, and the overall system has a mass of around 500 g and is approximately 10x10x4cm in size. INSPIRE would use the RAX-2 system, which would be sufficient for the L5 mission as well.

Navigation/Communication: While the SolWISE proposal made use of an S-Band radio, the INSPIRE mission would use a JPL X-band navigation/communication radio called Iris that is capable of coherent transmissions for radiometric tracking of the spacecraft. This radio allows for 1000 bps at 1.5 Mkm distances to a Deep Space Network 34 m dish using INSPIRE's onboard omnidirectional patch antenna at 5W of radiated power. At 0.5 kg and 10x10x4 cm, the Iris radio has heritage from the Low-Mass Radio-Science Transponder (LMRST) and NASA's Electra radio. For this mission to L_5, the same Iris-based communication/navigation system, but using a ~0.5m–diameter dish antenna, would be sufficient for inter-spacecraft communication, sending engineering data directly to Earth, and for radiometric tracking of each constellation member. The inter-spacecraft communication rate would roughly 2Mbps, assuming a 5W transmitter and 0.5m antenna. One such 0.5m deployable dish antenna is currently flying on the University of Southern California's *Aeneas* CubeSat; it deploys much like an umbrella (see http://www.isi.edu/projects/serc/aeneas).

Spacecraft 1, the communication hub, would need a larger antenna and more radiated power. Based on 70+W solar panels now being offered for CubeSats and the continuous solar illumination for this mission, the transmitter dc input power available would be >20W continuous or >50W in burst mode. It should be possible to fit an ~1m deployable antenna and the more powerful communication systems in the ~2U additional space allocated for this purpose in Spacecraft 1. CubeSat navigation/communications subsystems are evolving rapidly and the L_5 mission could take advantages of these advances.

Attitude Determination and Control System (ADCS): Today's attitude determination and control systems often make use of magnetometers and magneto-torquers to orient the spacecraft in LEO. Both of these systems become inoperable in deep space due to the lack of a strong enough magnetic field. Instead, for the L5 mission, a reaction wheel system would be needed to torque the solar sail into the correct position, and the solar sail itself and adjustable solar panels (as in the SolWISE concept, see Fig. 2) used to de-saturate the reaction wheels (using calibrated and controlled asymmetries in solar pressure). A system using cold gas thrusters could also be used as a backup for de-saturating the reaction wheels, for the initial despin of the spacecraft after deployment, and for performing any necessary small impulsive maneuvers. A three-dimensional printed cold-gas system, developed at the University of Texas for the Bevo-2 CubeSat (http://lightsey.ae.utexas.edu/research/bevo-2/), would be sufficient. The thruster system remains at approximately 10x10x5cm in size (~½ U), and weighs approximately 0.5kg. Integrated into this system, a small star tracker provides attitude determination accurate to less than 0.05°. A low-quality Micro-electromechanical systems (MEMS) gyroscope, along with basic photodiodes, allow for initial despin (by nulling rates and orienting toward the Sun). This system is based upon the INSPIRE attitude determination and control system, but expands its capability through the addition of reaction wheels per the SolWISE proposal. All the pieces needed for the SWB-L5 mission ADCS system exist and, after INSPIRE's flight, would have been tested in space, but they would not have been used together with the solar sail.

B. Science Instruments

The L_5 Heliospheric Imager: The required field-of-view for the L_5 heliospheric imager (L_5HI) is ~60° in order to image solar wind transients near the Sun-Earth line from around 3 solar radii out from the Sun (where most CMEs have formed well defined fronts) to Earth (see Fig. 1). A cadence of 30 mins to one hour would be sufficient to follow those fronts to Earth. The resolution and image size could be adjusted to fit the telemetry capabilities of a CubeSat (<1 kbps at present). The requirements are based on the actual operations of the Heliosphieric Imager (HI) instrument aboard the STEREO mission which has 1 kbps telemetry allocated to it. Images at much lower bit rate (~100 bps) are transmitted daily and used for operational purposes through a so-called space weather beacon (of 500 bps total downlink capability).

Thanks to the STEREO mission, the L_5HI has a strong design heritage. It derives from the designs of the heliospheric imagers on the Solar Orbiter (SoloHI) and Solar Probe Plus (WISPR) missions currently under development. WISPR, in particular, measures about 30cm x 14cm x 58cm and weighs 6 kg and is a smaller version of STEREO's HI instrument (55cm x 26cm x 84cm, and weighing 15 kg) currently in operation on the STEREO mission. WISPR is an optimal starting point for the L_5HI because (1) its focal plane is actually 1U, (2) it could easily accommodate a 60° telephoto lens, and (3) the instrument electronics meet our requirements in a ½ U package.

The most important factor for an HI telescope is successful suppression of the stray light entering the detector assembly. This requires baffling which defines the instrument size and can be quite complicated depending on the location of stray light sources on the spacecraft. In the case of a CubeSats, where L_5HI is the only instrument, the necessity for baffling is reduced, and the plane of its aperture is placed entirely sunward of the solar sail. Solar photovoltaic panels would need to be placed amidships, rather than on the sunward end of the spacecraft as shown for SolWISE, but this is a reasonably straightforward change. The width and height of the instrument are then driven by the size of the detector focal plane. The length of the instrument, on the other hand, is driven by solar occultation considerations. There needs to be a certain distance between the forward occulters and the lens depending on the desired minimum elongation angle from the solar limb. The longer the distance, the higher resolution and single-to-noise ratio we could achieve close to the solar limb.

To remain within the 2U volume, we insert the WISPR focal plane that is conveniently around 1U (10cm x 10cm) in the 2U box (see Fig. 6). To gain some extra distance to the occulters, we incorporate them on the top lid of the instrument, making it a deployable structure. This novelty would result to an effective 40cm distance when the lid deploys. This allows us to set 3-5 occulters along the first 10cm and thus minimize the solar stray light sufficiently to image starting as close as 3 solar radii from the Sun. We estimate the instrument mass at about 3 kg.

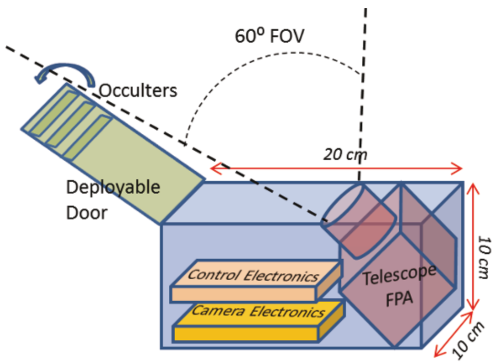

Fig. 6 Strawman concept for a 2U Heliospheric Imager based on the WISPR instrument on the Solar Probe Plus mission.

The telescope concept is based on technology at Technology Readiness Level (TRL) 6 and above. The deployable lid is a new concept. The structure must satisfy an alignment requirement between the last occulter and the lens of about <100 micrometers and must maintain the positioning of the occulters to the same accuracy. However, these requirements would not difficult to meet with properly designed and tested mechanical structures. Another open issue is the pointing stability of the spacecraft. Our concept of operations for the L5HI baselines a 2k x 2k CMOS/Active Pixel Sensor, developed for SoloHI, binned 2x2 to increase the signal-to-noise ratio. The effective spatial resolution is about 7 arcmin. This is beyond presently-demonstrated CubeSat pointing capabilities (of the order of one arcmin), and the instrument needs to maintain stable pointing for the duration of the exposure, which can be one minute or more. One approach being pursued for other investigations involves an MIT/JPL piezoelectric stage using guide stars, as described for the ExoplanetSat concept [10]. While development of pointing control mechanisms/algorithms to achieve arcmin-level pointing stability for long durations is required, at least one plausible path has been shown as referenced, and there are many applications driving the desire for CubeSat-compatible implementations.

The L_5 Magnetometer: Solar wind *in-situ* magnetic fields would be measured using a compact vector helium magnetometer (CVHM), which would occupy ~½ U volume at launch, and would have a boom-deployed sensor to reduce the effects of residual spacecraft magnetic fields. The CVHM is the latest in a long heritage of helium magnetometers, and is chosen because of its high stability, which facilitates accurate measurements of field differences between spacecraft. The CVHM is a laser-pumped helium instrument, where the traditional helium lamp is replaced with a diode laser. Laser pumping increases the potential sensitivity of the sensor significantly, with this sensitivity being traded for a reduction in volume in the CVHM. The elimination of the lamp reduces further the sensor size. The laser is fiber coupled to the sensor, allowing the laser diode and its electronics to be housed with the instrument electronics. The most recent heritage for the CVHM is the Dynamo sounding rocket experiment, which validated the fiber-coupled CVHM instrument. Further development is underway for JPL's INSPIRE CubeSat mission, with an anticipated launch as soon as 2014. The CVHM sensitivity would be < 10pT, with a frequency range of DC - 10 Hz. For the SWB-L5 mission, no further development is needed beyond what is being done now for INSPIRE. The data rate is variable with a peak at about 500 bps for10 samples/sec and 2 b/s at the low end for 2 samples/min; both rates assume a factor of two compression.

Solar Wind Instrument: The solar wind measurements required at L_5, the ambient velocity, density and temperature of the solar wind, can be measured by a Faraday Cup (FC) instrument. This could be packaged in 1U by taking advantage of modern electronics miniaturization. The single FC instrument derives its heritage from the instruments on NASA's WIND (Solar Wind Experiment-SWE) and on Triana [11, 12]. The instrument has a field of view of ~60° and provides a reduced velocity distribution of the solar wind along its mounting direction. It utilizes the spacecraft spin to determine the angular variation of the distribution in the spin phase direction, and a split pre-amplification and detection system from which the direction of the solar wind in the other direction (spin elevation). By on-board processing, moments of the distribution are thus obtained. The data rate would be approximately 50 bps, assuming a factor of two compression.

Energetic Particle Detector: The energetic particle detector is required at L_5 to determine, using the measured arrival times of protons and electrons accelerated at shocks, the shock distance at the time of the acceleration [13]. The arrival times of these particles, when correlated with X-ray fluxes as measured by NOAA Geostationary Operational Environmental Satellites (GOES) provide verification of the presence of reconnection or shock-acceleration at emerging shocks at the base of the corona. The approximate field line length as determined by modeling of local magnetic field measurements and remote observations of the photospheric field, could be used to ascertain the length estimates from energetic particle measurements. The requirements are energy range from 30keV to 500keV for ions and electrons, with an energy resolution dE/E ~ 50%, and an angular resolution of 45x45°. These could be easily met using the principles of solid state detector (SSD) systems flown on NASA's WIND, STEREO and Time History of Events and Macroscale Interactions during Substorms (THEMIS) missions, using standard electronics miniaturization technologies. The SSD instrument to be flown on Spacecraft 5, takes advantage of the spacecraft spin; it uses four detectors for each species with a combined 180° field of view along the elevation angle direction. This approach provides a full 4π steradian distribution once per spin with the aforementioned energy range and resolution, using a volume of 1U. The data rate would be approximately 500 bps, assuming a factor of two compression.

Magnetograph: The L_5 magnetograph (L_5MI) would provide line-of-sight magnetograms (magnetic field at the surface of the Sun) from the perspective of L_5, to allow early warning of the appearance of solar active regions before they become visible from Earth. Used together with magnetograms from Earth, observation from the L5 vantage point also enhances the fidelity of heliospheric solar wind models by providing a wider angular view of the photospheric magnetic field and thus a more accurate boundary condition on the magnetic field at the Sun. L_5MI would be a filter-based magnetograph, based on a magneto-optical filter (MOF) [14]. The MOF has extensive ground-based heritage in Doppler and magnetic imagers, and a flight version of the filter has been developed to TRL 6 at JPL. L_5MI would observe the photospheric Potassium line at 770nm, with a 5cm aperture. Recent flight designs of MOF based Doppler-magnetographs have estimated masses around 10 kg, for an instrument with a 7 cm aperture: we estimate that L_5MI, with it's reduced aperture would weigh approximately 5 kg, and could be packaged in a 2U volume. The data rate is 1200 b/s for a 6 hour cadence of 1024 by 1024 pixel magnetograms.

V. Summary and Discussion

In this paper, we have presented a concept for an operational space weather forecasting mission at L_5 using a constellation of five CubeSats. The concept presented could accomplish many of the goals of L_5 missions proposed using conventional spacecraft in the last decadal review, possibly at a much reduced cost. The constellation could later be expanded by adding other spacecraft with new science instruments and new goals. The fractioned Space Weather Base concept presented here represents a beginning for a permanent space weather base at L5. This is just one of the many advantages of this fractioned approach; this and others were discussed in Section I.

The orbits at L_5 would be achieved using a solar sail. The trajectory calculation results in Section II demonstrate that starting from an Earth Escape trajectory, it is feasible to send a constellation of multiple CubeSats using solar sail propulsion to reach and insert into a 1000 km diameter orbit around L_5 in less than three years. Starting from GEO altitude adds ~2.5 years to the travel time.

The very rapid rate of technology advances in the CubeSat community has made it so that progenitors of most of the science and engineering subsystems needed for this mission have already flown (RAX-2) or are ready to be flown (Bev-2). INSPIRE, to be launched to interplanetary space in ~2014, would fly several of them, including the communication and navigation system, the avionics system, the cold gas system and one science instrument, the magnetometer. It would also demonstrate inter-spacecraft communication. The other science instruments discussed in this paper also require little additional technology development.

For some of the technologies and subsystems, further technology development and/or flight testing is needed before this mission is ready to be executed. Certainly a mission is needed to test the solar sail system together with the attitude determination and control system (although all the necessary parts of this system exist). Also the existing avionics systems have not demonstrated the radiation tolerance (and thus lifetime) needed for this length of mission (at least 5 years). There is also a need to test the attitude control necessary for the heliospheric imager (spacecraft 2). As a precursor for this mission, a launch of two of the constellation members (the communication hub and spacecraft 2) to interplanetary space would give all the required flight tests. In addition, there is no design for the communication system for the constellation's communication hub, nor is there an operations concept. A more detailed design will be needed to fully encompass the mission requirements.

Even with this technology development and flight testing needed, it is clear that this mission could be launched before the next Heliophysics decadal review in ~2022. Looking further ahead, the mission concept presented here is on the path towards development of a solar sail/CubeSat-based "fractionated" Solar Polar Imager mission to explore the polar regions of the Sun [15].

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